PROBLEM NO. UDRI-1

Title: Crack Growth Analysis of Critical Area in Front Wing Spar and Verification of Model

Objective:

To illustrate the process of estimating crack growth behavior to set inspection limits and the process used to verify the analytical results.

General Description:

This problem focuses on a damage tolerance assessment of a critical area on a wing front spar for the purpose of establishing inspection intervals. The critical area includes both the spar cap and the wing skin. An airplane finite element model was developed to determine the stresses and the critical area was modeled using a standard AFGROW stress intensity factor solution. Verification testing was conducted to validate the life prediction model.

Topics Covered: Damage tolerance assessment, finite element analysis, crack growth analysis, inspection intervals

Type of Structure: wing skin, wing spar cap

Relevant Sections of Handbook: Sections 1, 2, 3, 4, 5, 7 and 11

Author: Peggy C. Miedlar

Company Name: University of Dayton Research Institute
Structural Integrity Division
Dayton, OH 45469-0120
937-229-4417
www.udri.udayton.edu

Contact Point: Peggy C. Miedlar
Phone: 937-229-4476
e-Mail: Miedlar@udri.udayton.edu

Overview of Problem Description

This problem focuses on a critical area on a wing front spar, shown in Figure UD-1.1 (photograph), and further described by the drawings of Figures UD-1.2 - UD-1.4. The critical area includes both the spar cap and the wing skin. The spar cap was fabricated from 2024-T3511 aluminum and the skin from 2024-T3 aluminum. The fasteners are 0.25 in diameter, and join the cap, skin and fitting. The specific area is shown in Figure UD-1.4, with the expected crack path marked.

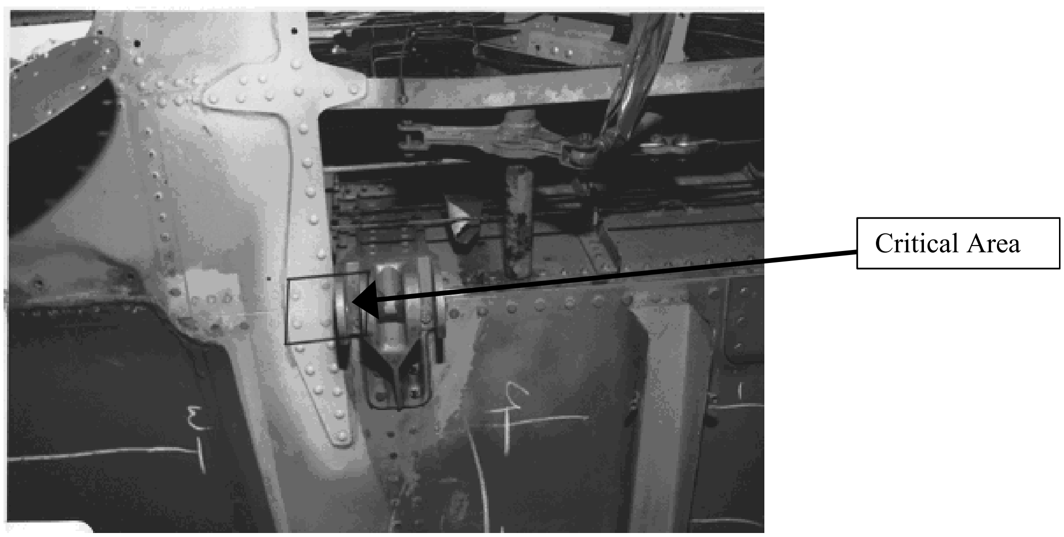


Figure UD-1.1. Photograph of Critical Area from Outside Wing.

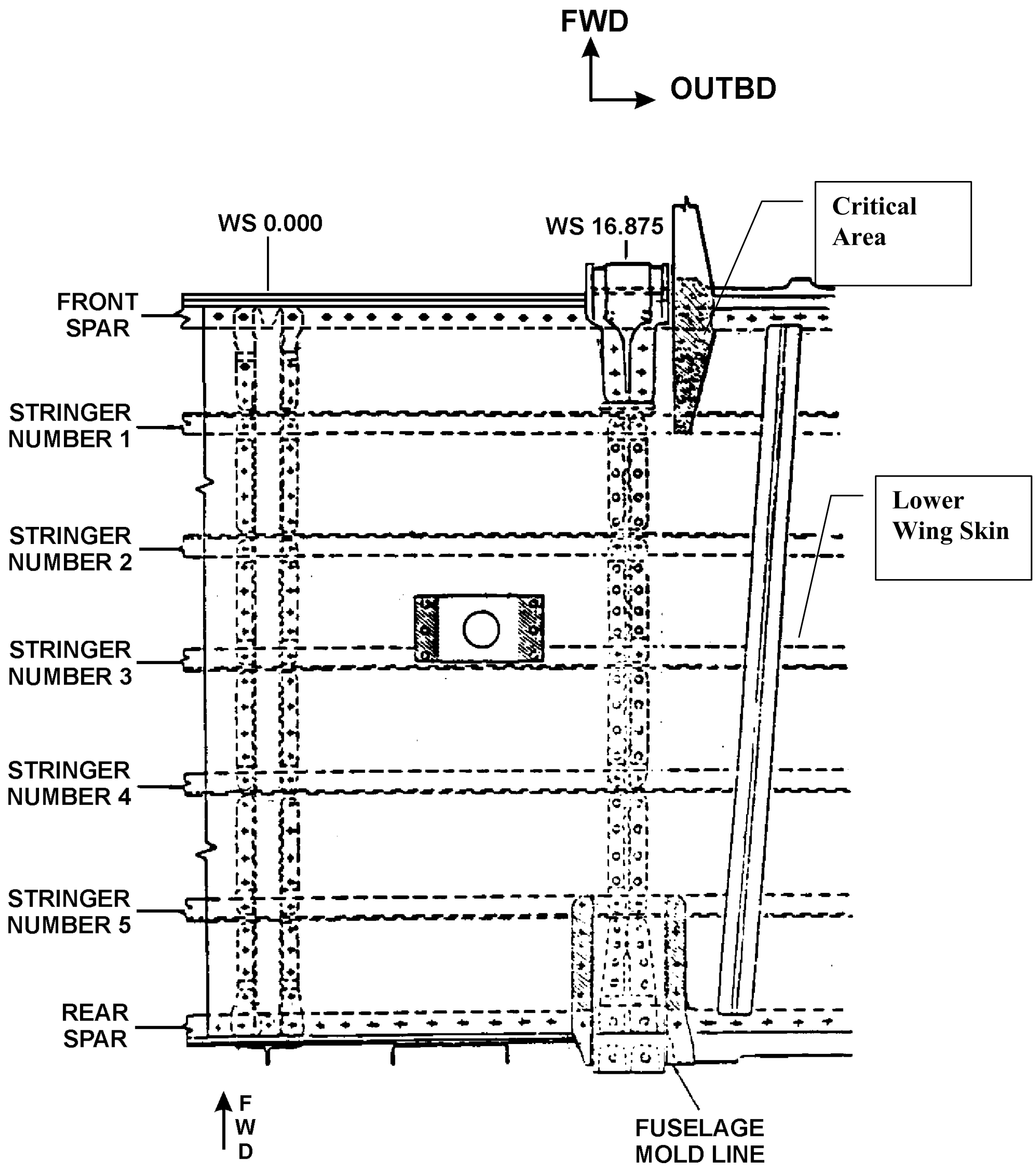


Figure UD-1.2. General Location of Critical Area.

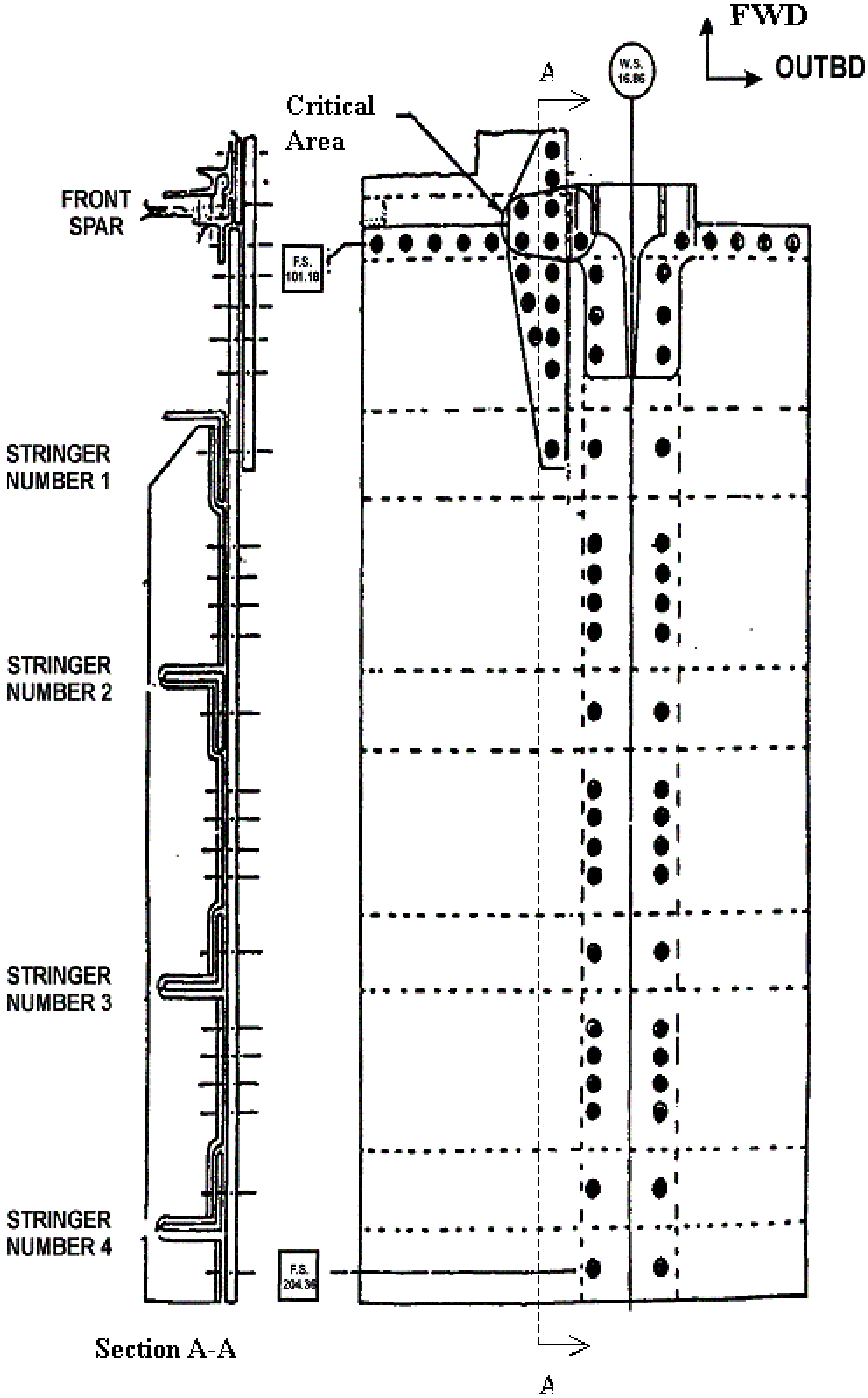


Figure UD-1.3. Structural Detail for Critical Area from Bottom of Wing

Looking Up.

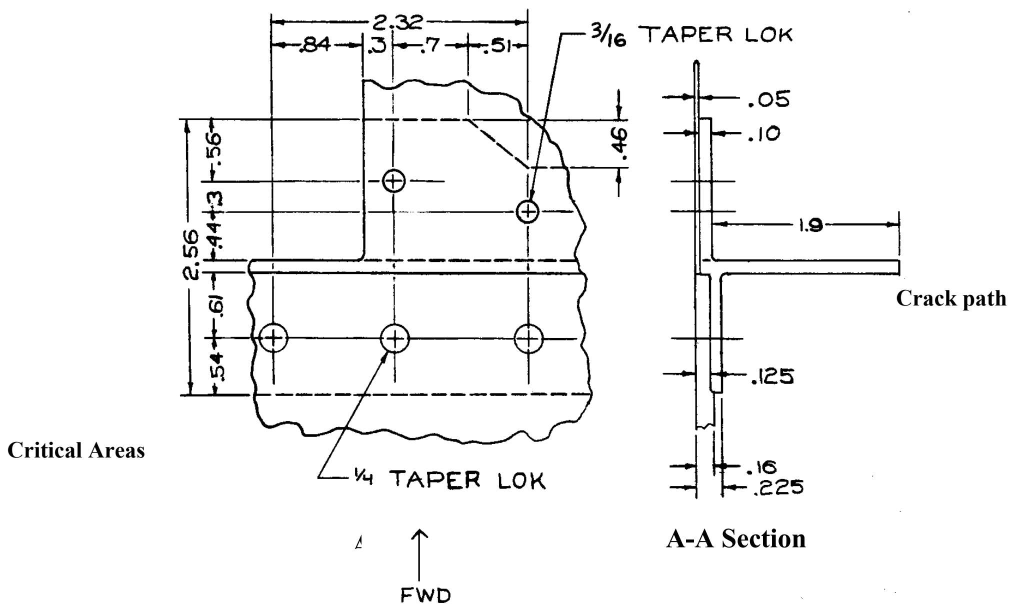


Figure UD-1.4. Detail Geometry of Critical Location Shown in Figure UD-1.3.

Structural Model

A finite element model was used for determining the level of stresses in the critical area. The loading for this geometry is tension.

Structural Model Details

If the details of the FE model are important to the problem, the model should be described here, using drawings to illustrate the model.

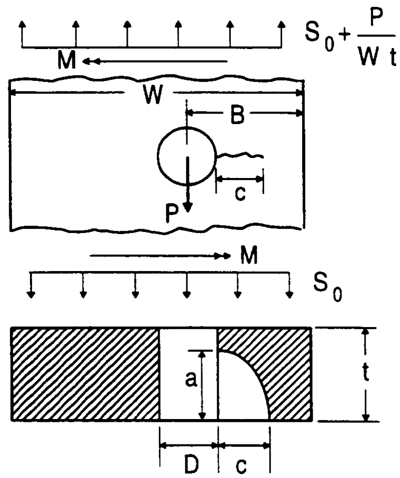
Model Geometry Description

The critical crack geometry was modeled as a corner crack from an off-centered hole, with the crack growing toward the short side. The corresponding AFGROW crack geometry model is called a single corner crack at a hole, as shown in Figure UD-1.5. A width (*W*) of 2.5 inches was assumed as representative of the distance from the plate edge to the next hole. The edge distance (*B*) is 0.61, thickness (*t*) is 0.125 and hole radius (*D*/2) is 0.125 inches.

Figure UD-1.6 describes the length direction beta factor (K/σ) for several *a/c* ratios.

Model Assumptions

Some assumptions were made for this analysis. Most of these assumptions are conservative, resulting in a shorter predicted life. These assumptions include: straight shank hole, open hole, no load transfer, no local residual stresses due to cold working, and no retardation.



W = 2.45 in.
B = 0.61 in.

Figure UD-1.5. Crack Geometry Model for Stress Intensity Factor.

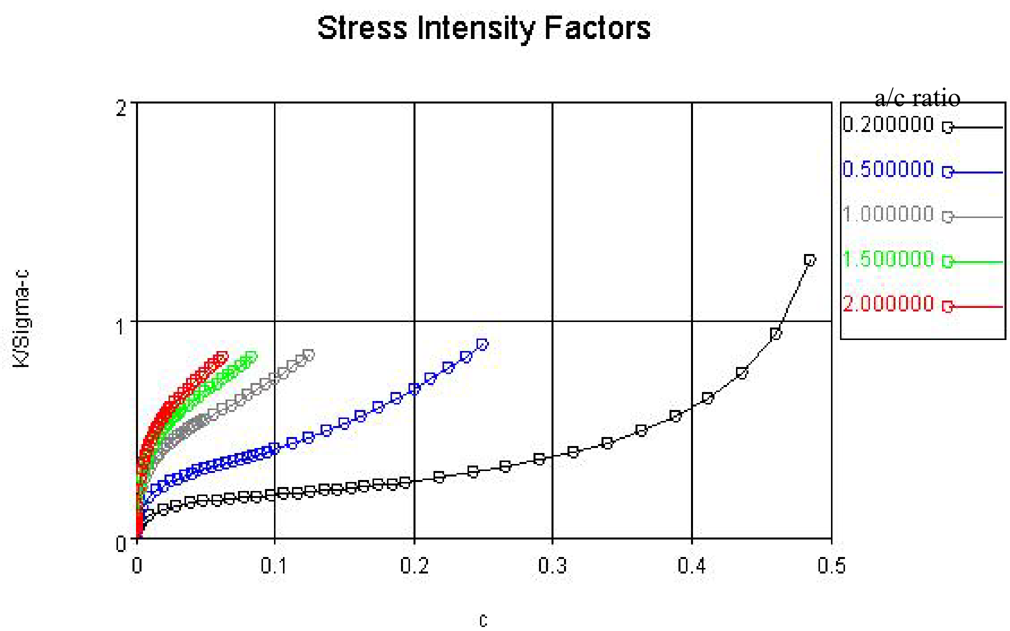


Figure UD-1.6. Surface Length Beta (K/σ) Factor for Corner Crack from a Hole for Several Different Crack Aspect Ratios (a/c).

Inspection Capabilities and Crack Limits

The holes in the flange and skin are covered by the wing-fuselage attachment fitting. With the fasteners removed, only the inside of the holes are visible. Therefore, these areas are inspected by X-ray. With X-ray inspection, the minimum detectable crack size in the field is 0.5 inch crack.

Structural Loading and Stress History Description

The stress spectrum is given in Table UD-1.1 where the flight history is presented as a fraction of the maximum spectrum stress (10.7 ksi). There are 1590 cycles in the spectrum, and this represents ten flights. Each flight is one hour.

Table UD-1.1. Flight History Data For Problem UDRI-1.

Step No.	Maximum Stress	Minimum Stress	Cycles
1	0.45	0.125	333
2	0.55	0.125	234
3	0.65	0.125	158
4	0.85	0.125	52
5	0.95	0.125	11
6	1.05	0.125	5
7	1.15	0.125	1
8	1.25	0.125	1
9	0.45	0.125	333
10	0.55	0.125	234
11	0.65	0.125	158
12	0.85	0.125	52
13	0.95	0.125	11
14	1.05	0.125	5
15	1.15	0.125	2

Material Property Description

The parameters for the Walker equation for the two aluminum alloys are given in Table UD-1.2, along with other material parameters. A detailed description as to how Walker constants were developed is presented in Section 5.

Table UD-1.2. Material Properties and Growth Rate Data.

Parameter	2024-T3	2024-T3511
Walker C	9.57×10^{-10}	9.57×10^{-10}
Walker n	3.7	3.7
Walker m	0.32	0.32
K_C	92.0	92.0
K_{IC}	35.0	46.0
σ_Y	48.0	54.0
ΔK_{th}	0.0	2.5

Solution Technique

This type of problem is conveniently solved using AFGROW. The input file for the AFGROW analysis is shown in Table UD-1.3.

Table UD-1.3. AFGROW Input File for Problem UDRI-1.

Data	Description
FAF012	
Example Problem	
~{Example problem using Walker equation and crack at off-centered hole }	Description of Problem
1030 0	
0.05	Geometry data
0.05 0	
0.05 0	
0.125	
2.5	
1 0 0 0	
0.25	
-1	
1 0.61	
10500	
0.33	
1.25e-005	
NOENVS	
NORETARD	
1	
1	
NOKMOD	
NOKRES	
WALKER_NEW	
2024-T3 example	Material Data
1	
9.57e-010 3.7 0.32	
92 2	
35 -0.3 0.99 48	
0	
NO_INITIATION	
10.7	
0	Spectrum data
SPFILE	
spectrum.sp3	

The spectrum is contained in a separate file named *spectrum.sp3*, and is shown in Table UD-1.4. Each repeat of the defined segment represents ten flights, and each flight represents one hour.

Using AFGROW terminology, the spectrum is entered as a blocked spectrum with one sub-spectrum. In this case, the sub-spectrum is the block of stresses given in Table UD-1.1.

Table UD-1.4. AFGROW Sub-Spectrum File for Problem UDRI-1.

Data	Description
1 15	Sub-spectrum number, number of levels
0.45 0.125 333	Maximum stress, minimum stress, number of cycles
0.55 0.125 234	
0.75 0.125 158	
0.85 0.125 52	
0.95 0.125 11	
1.05 0.125 5	
1.15 0.125 1	
1.25 0.125 1	
0.45 0.125 333	
0.55 0.125 234	
0.75 0.125 158	
0.85 0.125 52	
0.95 0.125 11	
1.05 0.125 5	
1.15 0.125 2	

Results

Critical crack size/Residual Strength

Using the Irwin Criterion for fracture, i.e.,

$$K_{\max} = \sigma\beta\sqrt{\pi c} = K_{Ic} = 92\ ksi\sqrt{in}$$

This criterion is imbedded in the AFGROW code and is used to determine the critical thru-thickness crack size (c) = 0.458 inches. The corner crack transitions into a thru-thickness crack at about one-half of the life.

Life:

Based on the calculations for growing the crack in AFGROW, the life from initial crack size to failure is determined to be 3100 hours. The results of crack length versus life and crack depth versus life are shown in Figures UD-1.7 and UD-1.8, respectively. The life is given in flight hours.

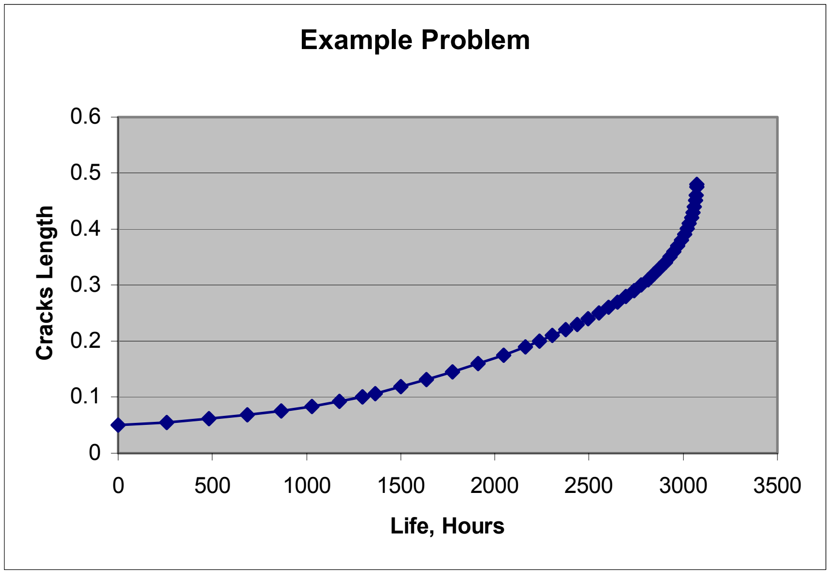


Figure UD-1.7. Crack Length versus Life for Problem UDRI-1.

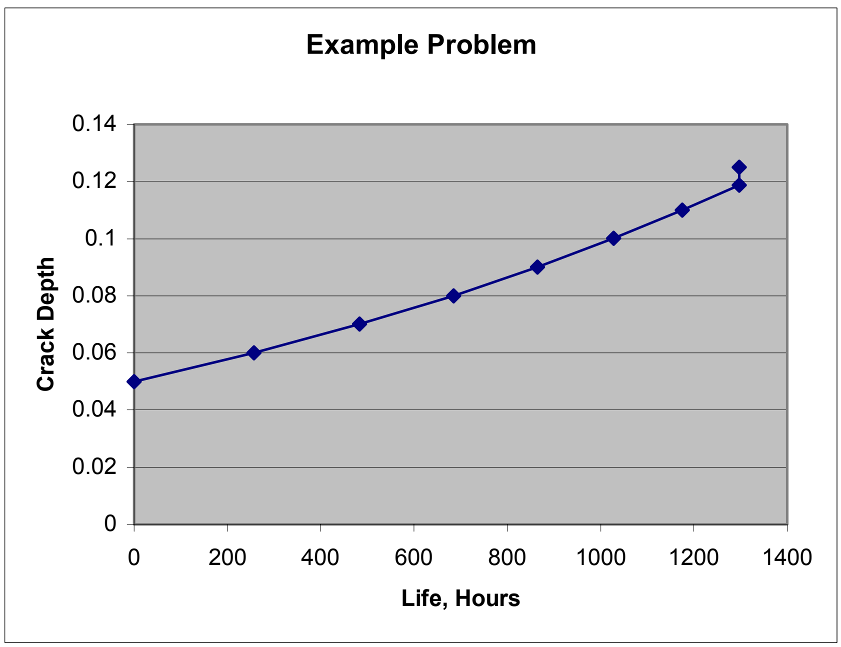


Figure UD-1.8. Crack Depth versus Life for Problem UDRI-1.

Verification of the Life Analysis

To verify the analysis procedure, four specimens were tested under the operational stress spectrum, and these results were compared to the analytical results. The test specimens were designed to represent the localized features that match the actual aircraft structure, seen in Figure UD-1.10. The operational spectrum is given in Table UD-1.1. The test results are summarized in Table UD-1.8.

The AFGROW program was used to predict the specimen lives. The results of the analysis are also shown in Table UD-1.8. The predicted results are compared to the analytical results with the ratio of predicted life divided by actual life (N_P/N_A).

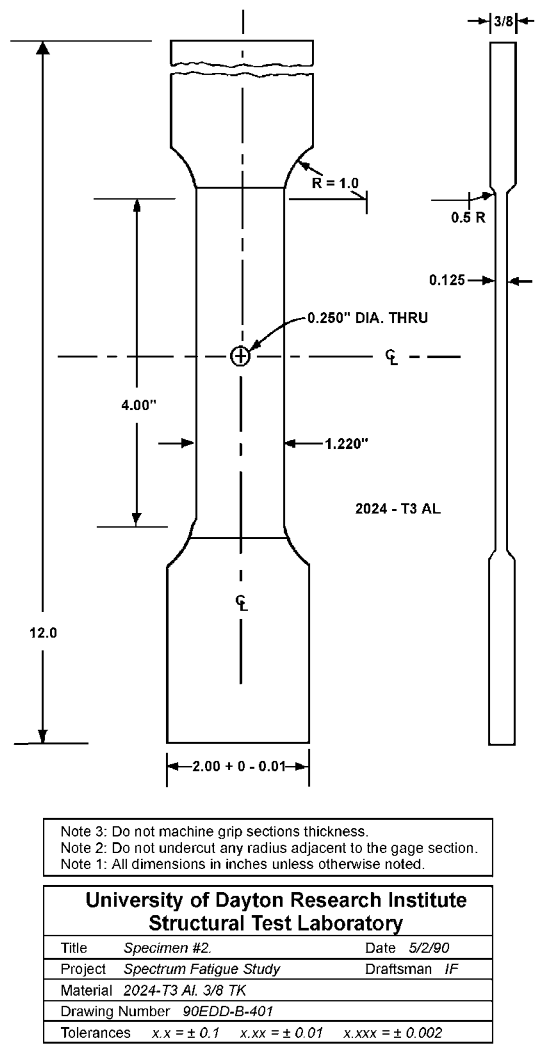


Figure UD-1.10. Test Specimen.

Table UD-1.8. Test Results for 2024-T351 C(T) Specimens.

Specimen ID	Width (in)	Thickness (in)	Hole Diameter (in)	Precrack Length (in)	Test Flights to Failure (N_A)	Predicted Flights to Failure (N_P)	$\frac{N_P}{N_A}$
5	1.220	0.123	0.250	0.050	2072	1616	0.78
8	1.221	0.124	0.249	0.050	1844	1697	0.92
11	1.220	0.124	0.250	0.049	1004	1626	1.62
12	1.220	0.125	0.250	0.048	2042	1736	0.85

Discussion of N_P/N_A

If the N_P/N_A ratio is equal to 1, then the analysis predicts the actual test results. If the N_P/N_A ratio is greater than 1, the analysis is unconservative. If the predicted life is less than the actual ($N_P/N_A < 1$), the analysis is conservative.

If the ratio is too high or low, i.e. $N_P/N_A=2$ or $N_P/N_A=0.5$, then the analysis method and assumptions should be reviewed to rectify the differences between the experiment and analysis

For these tests, the N_P/N_A ratios show a good correlation between the test results and analysis. Three of the four tests show that the analysis is conservative.

Inspection Intervals

The initial inspection interval is at one-half of the life. For the predicted life of 3100 hours, the first inspection is set at 1550 hours.

Subsequent inspections are one half the life from NDI field detectable crack size to the critical crack size. However, for this problem, the failure occurs prior to the field detectable crack size.

Force Management Decisions

Since the critical crack size (a_f)= 0.458 inch is less than the NDE detectable size (a_{NDE} = 0.5 inch), the situation precludes the use of multiple inspections. And the structure must be classified as slow crack growth critical. This means that once the initial inspection period has been reached, the life limit of the structure has been reached, i.e., the life is 1550 flights (=3100/2). Alternately, one could use the results to assess different inspection and repair options. For example, if an inspection method can be found that will detect the presence of 0.005 inch long cracks, then the time between inspections becomes 6575 flights. Thus, if after pulling the fasteners from the holes for the first in-depth inspection, these holes are then coldworked, the lives can be extended tremendously, and subsequent inspections might not be required.

Complementary Sensitivity Studies

- Cold working of holes/compressive residual stresses due to taper-lok.
- Filled hole load transfer.

- Taper-lok holes – one method for accounting for fatigue rated fasteners systems is to start the analysis with initial crack size of 0.005 inches.
- Retardation model (currently using no retardation).